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EFFECT OF MACH AND REYNOLDS NUMBERS ON THE MAXIMUM LIFT

COEFFICIENT OBTAINABLE IN GRADUAL AND ABRUPT STALLS

OF A PURSUIT AIRPLANE EQUIPPED WITH A LOW-DRAG WING

By John R. Spreiter, George M. Galster, and William K. Blair

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Moffett Field, California

WASHINGTON

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

MEMORANDUM REPORT

for the

Bureau of Aeronautics, Navy Department

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SUMMARY

Flight tests were conducted on a pursuit airplane, which has an
NACA low-drag wing, to determine the effects of Mach and Reynolds
numbers on the maximum lift coefficient obtainable in gradual and
abrupt stalls. Gradual stalls were made at Mach numbers from 0.145
to 0.67 and Reynolds numbers from 5,200,000 to 19,300,000. Stalls
of varying degrees of abruptness were made at selected Mach numbers
from 0.195 to 0.44 and Reynolds numbers from 6,370,000 to 11,300,000.

The test results indicated that the maximum lift coefficient
obtainable in a gradual stall was greatly affected by Mach number as
well as Reynolds number, even when the Mach number was as low as 0.15.
As the Mach number was increased from the lowest value tested, the
maximum lift coefficient decreased steadily until a minimum value of
0.90 was reached at a Mach number of 0.49 and then increased reaching
a value of 1.09 at a Mach number of 0.66. In addition, the usual
Reynolds number effects were postponed to a larger Reynolds number
and were diminished in magnitude until, at speeds greater than the
critical Mach number, no effects of Reynolds number were apparent.
The maximum lift coefficients obtainable in abrupt stalls were found
to be limited by Mach number but were independent of Reynolds number.

A comparison is made with corresponding data obtained for a
pursuit airplane equipped with a conventional wing, which is very
similar to the test airplane. Results show that at low Mach numbers
the maximum lift of the airplane with the conventional wing was
greater than that of the airplane equipped with the low-drag wing.
At moderately supercritical Mach numbers, however, the maximum lift
of the airplane with the low-drag wing was much greater than that of
the airplane with the conventional wing.

INTRODUCTION

While extensive information on the characteristics of NACA low-drag airfoils has been provided by wind-tunnel tests, the characteristics of practical construction wings having the low-drag airfoil sections have not been as completely evaluated. In order to provide comparative data on the maximum lift characteristics of wings having low-drag and conventional airfoil sections, flight tests have been conducted at the Ames Aeronautical Laboratory on two airplanes of otherwise similar configuration.

Reference 1 presents data on the effects of Mach and Reynolds numbers on the maximum lift coefficient obtainable in gradual stalls of a pursuit airplane which has a wing with conventional airfoil sections. In the present report information is provided on the effects of Mach and Reynolds numbers on the maximum lift coefficient obtainable in gradual and abrupt stalls of a similar pursuit airplane which has a wing with low-drag airfoil sections. For purposes of comparison data from reference 1 are included in the present report.

DESCRIPTION OF THE AIRPLANE

The test airplane is a single-place, single-engine, low-wing, cantilever monoplane. Figure 1 is a three-view drawing of the airplane and figure 2 shows the airplane as instrumented during the tests. The profiles of the root and tip airfoils are shown in figure 3. The general specifications of the airplane are as follows:

Engine . . . . Liquid-cooled, V-12, V-1710-93

Propeller. Hydraulically operated, constant speed

   Diameter. . . . . 11 ft 7 in.

   Propeller design. A642-S-D1

   Blade design. A-20-156-17

Weight at take-off (as flown). . . . . 8300 lb

Center-of-gravity position at take-off
   (as flown). . . . . 26.3 percent M.A.C.

Wing

   Span. . . . . 38.33 ft

   Area. . . . . 248 sq ft

Aspect ratio. 5.93

Taper ratio 2:1

Incidence, root . . . . $1^{\circ}18'$

Incidence, tip. . . . . $-0^{\circ}27'$

Dihedral (top surface, 35 percent chord). . . . . $3^{\circ}40'$

Sweepback (leading edge). . . . . $5^{\circ}6'17''$

M.A.C. . . . . 6.88 ft

Airfoil, root . . . . NACA 66,2X-116(a=0.6)

Airfoil, tip. . . . . NACA 66,2X-216(a=0.6)

Horizontal–tail surfaces

Incidence . . . . 1°

Total area . . . . 45.15 sq ft

INSTRUMENTATION

Standard NACA photographically recording flight instruments
were used to measure, as a function of time, the following variables:
indicated airspeed, pressure altitude, normal acceleration, and
pitching and rolling velocities. The head used for the measurement
of airspeed was mounted on a boom extending 4.2 feet ahead of the
leading edge and located 2.1 feet inboard of the left wing tip. The
installation was calibrated for position error. Indicated airspeed
as used in this report is defined by the following formula by which
standard airspeed meters are calibrated:

$$V_i = 1703 \left[\left(\frac{H-p}{p_0} + 1 \right)^{0.286} - 1 \right]^{\frac{1}{2}}$$

where

V_i correct indicated airspeed, miles per hour

H free-stream total pressure

p free-stream static pressure

p_0 standard atmospheric pressure at sea level

The free-air temperatures were obtained from radiosonde obser-
vations taken during the mornings and evenings of the dates of the
test flights and were checked in flight by reading a free-air-
temperature indicator connected to a temperature bulb mounted under
the wing of the airplane.

TEST PROCEDURE

Tests were made with the flaps and gear up, oil- and coolant-
duct shutters closed, power off, and with the propeller in the
constant-speed high-pitch position.

Gradual stalls were made in turns at speeds throughout the
available range at 5,100, 8,800, 13,500, 17,500, 21,300, 24,500,
29,400, and 32,300 feet pressure altitude. Stalls of varying degrees
of abruptness were made at selected speeds and altitudes throughout
the available Mach and Reynolds number range. With a few exceptions,
all stalls were made within 300 feet of the listed altitude and 3
miles per hour of the listed speeds.

RESULTS AND DISCUSSION

The maximum lift coefficient obtainable by an airplane in
flight may be limited by uncontrolled-for motions or very severe
buffeting. This point is emphasized because the maximum lift coef-
ficient obtainable in flight is determined not only by the maximum
lift coefficient as measured in a wind tunnel, but also by the con-
trollability when near the stall. The test airplane usually pitched
down and rolled somewhat at the stall. At the highest Mach numbers
tested, however, the airplane rolled suddenly to the right at the
stall. These control characteristics at the stall may have prevented
the attainment of the ultimate maximum lift coefficient that would
be measured in a wind tunnel. Buffeting severe enough to prevent
the maximum lift coefficient from being attained was not encountered.

In computing lift coefficients, the lift was assumed equal to
the normal force WA_Z; it was estimated that the error involved in
this assumption was less than 3 percent.

$$C_{L_{max}} = \frac{WA_Z}{qS}$$

where

$C_{L_{max}}$ maximum lift coefficient

W   weight of the airplane, pounds

A_Z normal acceleration factor, the ratio of the net
         aerodynamic force along the airplane Z-axis at the stall
         (positive when directed upward), to the weight of the
         airplane

S   wing area, square feet

q   dynamic pressure at the stall, pounds per square foot

Gradual Stalls

The maximum lift coefficient obtainable in gradual stalls is
the quantity most similar to the steady-state maximum lift coeffi-
cient measured in wind tunnels. As such, it represents the maximum
useful lift coefficient and is of importance in estimating the
landing speed and the turning ability of an airplane.

Curves of the maximum normal acceleration factor obtained in
gradual stalls, corrected to an airplane weight of 8000 pounds, are
plotted as a function of indicated airspeed at each test altitude in
figure 4 and are cross-plotted as a function of altitude for constant
indicated airspeed in figure 5. These curves show that, at low
speeds, the maximum normal acceleration factor decreases with an
increase in altitude. At high speeds and altitudes, however, a
critical point is reached beyond which the maximum normal accelera-
tion factor increases with an increase in altitude. The reasons for
these variations will be shown in the subsequent sections of the
report.

Effect of Mach number on the maximum lift coefficient.— The
variation with Mach number of the maximum lift coefficient obtainable
in gradual stalls at each test altitude is shown in figure 6. At
each altitude the maximum lift coefficient decreases to 0.90 as the
Mach number increases to 0.49 and then increases with further
increases of Mach number reaching a value of 1.09 at a Mach number
of approximately 0.66. There is an indication of a tendency to peak
at this value, which is similar to the tendency previously observed
in wind-tunnel tests. (See reference 2.) In the low Mach number
region, the maximum lift coefficient decreases markedly with increases
of altitude; whereas at higher Mach numbers the effects of altitude
become very small.

The data shown in figure 6 are replotted in figure 7 to show
the variation of the maximum lift coefficient with Mach number for
constant Reynolds numbers. For all except the 6,500,000 and
8,000,000 Reynolds number lines the maximum lift coefficient
decreases almost linearly with increases of Mach number M at a
rate $\dfrac{\delta C_{L_{max}}}{\delta M}$ of approximately −1.75. This variation may be caused,

as described in reference 2, by the thickening of the boundary layer
which reduces the circulation or by the earlier separation resulting
from the steeper adverse pressure gradients associated with larger
Mach numbers.

Theoretical computations made in reference 3 indicated that the
wing tip and root airfoil sections have critical Mach numbers of 0.44
and 0.40, respectively, at a lift coefficient of 0.90. It appears,
therefore, that the increase in the maximum lift coefficient is
caused by an increase in the chordwise extent of the low-pressure
region on the upper surface following the formation of a supersonic
velocity region. This phenomenon has previously been noted in
reference 2. In order to illustrate the point, pressure distribu-
tions from reference 2 for an NACA 16-515 airfoil at an angle of
attack of 11° at Mach numbers of 0.40, 0.55, and 0.60 are presented
in figure 8. These pressure distributions show that, as the speed
is increased from subcritical speeds to moderately supercritical
speeds, the upper-surface pressure-distribution changes from one
with a sharp negative pressure peak followed immediately by a steep
adverse pressure gradient to one with a lower, more rounded, negative
pressure peak with the steep adverse-pressure-gradient region of the
shock wave moving rearward with increases of Mach number. While this
phenomenon is occurring on the upper surface, the lower-surface
pressure distribution is remaining essentially unchanged, thereby
accounting for the gain in lift. Unpublished data on file at Ames
Aeronautical Laboratory show that similar changes in pressure distri-
bution occur on an NACA 66,2-215 airfoil which is very similar to
that used on the test airplane. As the critical Mach number is first
exceeded, the rounded pressure peak produces a higher maximum lift
coefficient, but as the Mach number is increased further, the loss in
lift due to the decreasing values of the limit negative pressure
coefficients (a concept presented in reference 4) finally overcomes
the increase due to the rearward movement of the shock wave.

Figure 9 presents data obtained from reference 1 showing the
variation of the maximum lift coefficient with Mach number for
constant Reynolds numbers for power-off gradual stalls of the air-
plane equipped with the conventional wing. This airplane is similar
to the test airplane except for the wing sections. The wing of the
airplane described in reference 1 consists of NACA conventional
sections tapering from an NACA 0015 at the wing root to an NACA 23009
at the tip. It should be noted that the curves for the airplane with
the conventional wing are slightly different from those originally
presented in reference 1, due to the correction of some small errors.
A comparison of the curves of figure 9 with the corresponding data
for the test airplane reveals that the character of the variation of
the maximum lift coefficient with Mach number differs greatly for
the two airplanes. In contrast to the variations previously noted
for the test airplane, the maximum lift coefficient of the airplane
of reference 1 decreased steadily with increasing of Mach number
throughout the entire Mach number range tested. Unpublished airfoil

data on file at Ames Aeronautical Laboratory indicate that this char-
acteristic is typical of conventional airfoils and that the large
chordwise extension of the low-pressure region at moderately super-
critical Mach numbers illustrated in figure 8 does not occur on con-
ventional airfoils.

A comparison of the data obtained at low Mach numbers reveals
that the maximum lift coefficient of the airplane with the conven-
tional wing is greater than that of the test airplane. At super-
critical Mach numbers, however, the maximum lift of the conventional
wing was much less than that of the low-drag wing.

Effect of Reynolds number on the maximum lift coefficient.—
Figure 10, a cross plot of figure 7, shows that as the Reynolds
number is increased at constant Mach numbers the maximum lift coef-
ficient at first remains nearly constant; but when a critical
Reynolds number is reached, the maximum lift coefficient increases
rapidly to a higher value and then remains nearly constant again
with further increases of Reynolds number. The rapid increase of
the maximum lift coefficient has been shown in reference 5 to be
associated with the change from laminar separation to turbulent
separation. Figure 11 shows that the critical Reynolds number
(Reynolds number at which the maximum lift coefficient starts its
rapid increase) increases nearly linearly with Mach number. There
are two effects which could cause such a variation of critical
Reynolds number with Mach number. One is the increase of kinematic
viscosity in the boundary layer of a compressible fluid due to aero-
dynamic heat which causes the ratio of the local Reynolds number
(based on boundary-layer conditions) to the free-stream Reynolds
number to diminish as shown in reference 6. Accordingly, as shown by
references 7 and 8, a larger free-stream Reynolds number would be
necessary to reach the local critical Reynolds numbers required for
transition from a laminar to a turbulent boundary layer. A second
possibility is due to the fact that increasing the Mach number in the
subcritical range has effects on the upper-surface pressure distribu-
tion similar to that of decreasing the airfoil thickness. The pres-
sure peaks become sharper and the adverse pressure gradients become
steeper. Decreasing the airfoil thickness is shown in reference 5 to
increase the critical Reynolds number, hence similar effects due to
increasing Mach number would be anticipated.

As the Mach number increases, the effects of Reynolds number on
the maximum lift coefficient decrease until, at supercritical Mach
numbers, no effects are apparent. The Reynolds number effects are
probably suppressed in the supercritical Mach number region because
the separation point may be determined entirely by the position of
the shock wave rather than by the gradual growth of the boundary
layer that occurs in subcritical flow.

Figure 12 for the airplane of reference 1 indicates that the
Reynolds number effects on the maximum lift coefficient are of a

similar nature to those determined for the test airplane. As the
Mach number is increased, the effects of Reynolds number decrease
until at a Mach number of 0.60, no effects are discernible. Although
not included in the test range, it appears from the shape of the
curves that the critical Reynolds number increases with increases of
Mach number in a manner similar to that observed for the test air-
plane.

The results of these tests indicate the necessity for consider-
ing both the individual and interrelated effects of Mach and Reynolds
numbers in the prediction from wind-tunnel-model data of the maximum
lift coefficient of an airplane flying at subcritical Mach numbers.
This becomes of special importance in the prediction of landing
speed. At speeds greater than the critical Mach number, however, the
maximum lift coefficient is shown to be independent of Reynolds
number over the range of the tests (9,000,000 to 16,500,000).

Calculated minimum radius of turn.- The effects of the varia-
tion of the maximum lift coefficient with Mach and Reynolds numbers
on the maneuvering characteristics of the test airplane are illus-
trated by the curves of figure 15 showing the calculated minimum
radius of a properly banked, power-off, horizontal turn at various
altitudes plotted as a function of indicated airspeed and that com-
puted by using a low Mach number value of the maximum lift coeffi-
cient (1.30).

A comparison of the two curves for each altitude shows the large
detrimental effects on the turning ability of the test airplane that
result from the decrease of the maximum lift coefficient with
increases in Mach number. This comparison indicates the necessity
for considering Mach number effects in estimating the turning per-
formance of airplanes, especially at high altitudes.

Abrupt Stalls

Although the higher lift obtained in abrupt stalls is not a
useful quantity in increasing the maneuverability or decreasing the
landing speed of an airplane, it is considered, at present, in
designing airplanes to withstand loads imposed by vertical gusts and
by abrupt maneuvers. Although the tests were made by stalling the
airplane in very abrupt pull-ups, similar maximum lift characteris-
tics would be anticipated in stalls produced by flying into a verti-
cal gust because both phenomena are caused essentially by the lag of
the flow separation following a sudden increase in the angle of
attack. The effects of pitching velocity and Mach and Reynolds
numbers on the maximum lift coefficient will be discussed in the
following sections.

Effects of pitching velocity on the maximum lift coefficient.-
The effects of pitching velocity on the maximum lift coefficient are

shown in figure 14 where the maximum lift coefficient is plotted for various speeds and altitudes as a function of the excess pitching angle per chord length of travel. This parameter is

$$\frac{\Delta q \bar{c}}{V_T}$$

where

Δq excess pitching velocity (the difference between the actual pitching velocity and that required to maintain the airplane in a steady turn or pull-up at a constant angle of attack corresponding to the speed, altitude, normal acceleration, and attitude of the airplane at the time of the stall), degrees per second

$\bar{c}$ mean aerodynamic chord, feet

V_T true airspeed at the time of the stall, feet per second

Use of this parameter to aid in the general application of these data is justified in reference 9 where it is pointed out that it represents the condition for dynamic similitude of unsteady acceleration forces.

Figure 14 shows that the maximum lift coefficient increases nearly linearly with rate of pitching until a limiting value of the maximum lift coefficient is reached which is unaffected by further increases in pitching velocity.

Effect of Mach and Reynolds numbers on the maximum lift coefficient.- The limiting value of the maximum lift coefficient obtainable in abrupt stalls is plotted as a function of Mach number and altitude in figure 6 and is shown to decrease rapidly with increases of Mach number but to be independent of the altitude and, consequently, the Reynolds number. As a result of theoretical calculations indicating that the critical Mach number of the wing is exceeded at lift coefficients below the limiting value obtained in the abrupt stalls, it appears that the maximum lift coefficient is limited by the limit pressure coefficients or by boundary-layer separation induced by the compression shock wave rather than by the normal growth and separation of the boundary layer. The maximum lift coefficient, therefore, becomes relatively independent of the lag of the flow separation and remains nearly constant with further increases of pitching velocity.

The trends of the curves shown in figure 6 suggest that the maximum lift coefficient obtainable in abrupt stalls may approach that obtainable in gradual stalls at a Mach number of approximately 0.66. Such an extrapolation appears plausible because, in both gradual and abrupt stalls at these Mach numbers, the maximum lift coefficient is becoming limited by the reduction in limit pressure coefficient.

Comparison With Calculated Limit Lift Coefficient

The limit lift coefficient, a concept introduced in reference 4, is defined as the lift coefficient at which potential flow ceases to exist (viz, the lift coefficient at the so-called compressibility burble) and is shown to be in agreement with the maximum lift coefficients measured in abrupt stalls of a P-47C-1 airplane.

The limit lift coefficient was computed for the test airplane by the method outlined in reference 4. Figure 15 shows a comparison of this calculated limit lift coefficient with the maximum lift coefficient obtainable in the gradual and abrupt stalls. At Mach numbers less than 0.61, the curve shows that the calculated limit lift coefficient is less than the maximum lift coefficient measured in abrupt stalls, and that this limit lift coefficient decreases with increases in Mach number. At a Mach number of 0.61, however, the calculated limit lift coefficient increases suddenly, reaching, at a Mach number of 0.64, a value slightly greater than the maximum lift coefficient measured in gradual stalls at that Mach number. With further increases in Mach number, the limit lift coefficient decreases abruptly reaching a value of zero at a Mach number of approximately 0.69.

Although the value of the limit lift coefficient is in fair agreement with the maximum lift coefficient at Mach numbers less than 0.65, the temporary rise in the limit lift coefficient at a Mach number of 0.61 is merely a coincidence; and the abrupt drop at a Mach number of 0.64 is not indicated by the trend of the flight data, nor is it shown by unpublished wind-tunnel data (on file at Ames Aeronautical Laboratory) on similar sections tested to much higher Mach numbers. The discrepancy is traceable, in this instance, to rather large differences between experimental high Mach number pressure distribution and those predicted by the method of reference 4. The sudden increase of the calculated limit lift coefficient at a Mach number of 0.61 occurs when the angle of attack decreases to the value at which the minimum-pressure-peak position moves rapidly from near the leading edge to the 60-percent-chord station. With further increases of Mach number, the negative lift on the lower surface increases rapidly until, at a Mach number of 0.685, the calculated limit lift coefficient approaches zero or an indeterminate value.

CONCLUSIONS

From tests of the maximum lift obtainable in gradual and abrupt stalls of the test airplane, the following conclusions have been made:

1. A limiting value of the maximum lift coefficient obtainable in very abrupt stalls was found which decreased with increases in Mach number and was independent of Reynolds number.

2. Reynolds number had less effect on the maximum lift coeffi-
cient obtainable in gradual stalls at high Mach numbers than at low
Mach numbers, no effects being apparent at Mach numbers greater than
0.50.

3. The increase in the maximum lift coefficient due to Reynolds
number occurred at higher values of Reynolds number at high Mach
numbers than at low Mach numbers.

4. The maximum lift coefficient was affected by compressibility
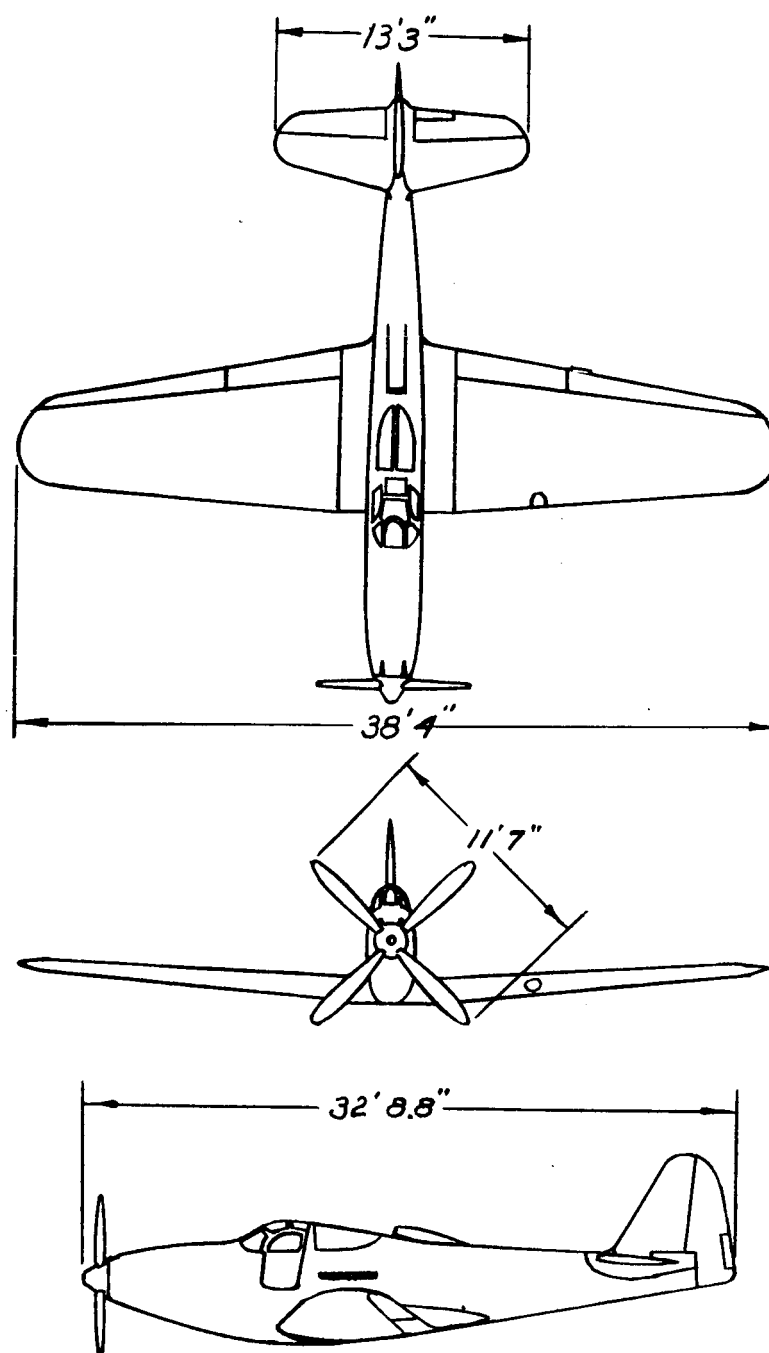at Mach numbers as low as 0.15.

5. The maximum lift coefficient obtainable in gradual stalls
decreased nearly linearly with increases in Mach number until a
minimum value of 0.90 was reached at a Mach number of 0.49, and then
increased with further increases in Mach number until a value of
1.09 was reached at a Mach number of 0.66.

6. At low Mach numbers the maximum lift of the airplane with
the conventional wing was greater than that of the test airplane
equipped with the low-drag wing. At moderately supercritical Mach
numbers, however, the maximum lift of the low-drag wing was much
greater than that of the conventional wing.

Ames Aeronautical Laboratory,
        National Advisory Committee for Aeronautics,
                Moffett Field, Calif., July 6, 1945.

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13'3"

38'4"

11'7"

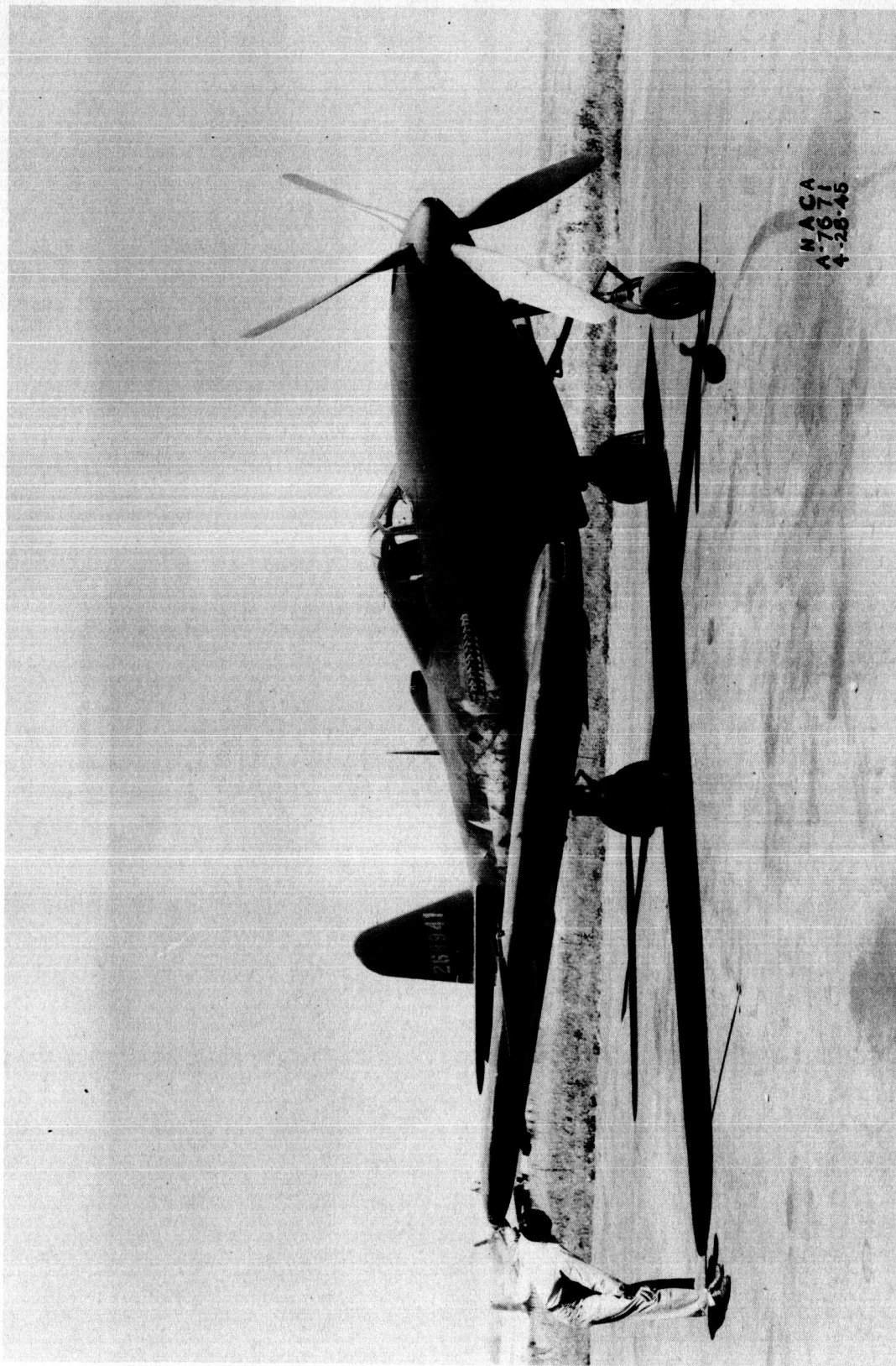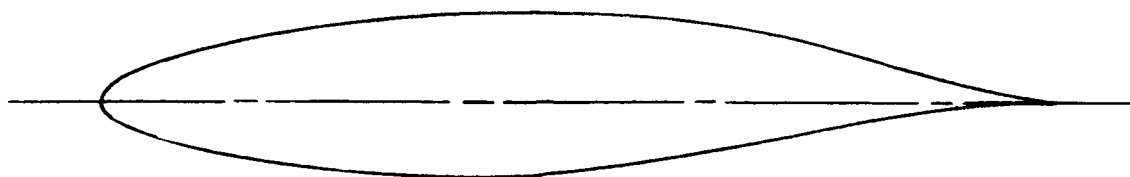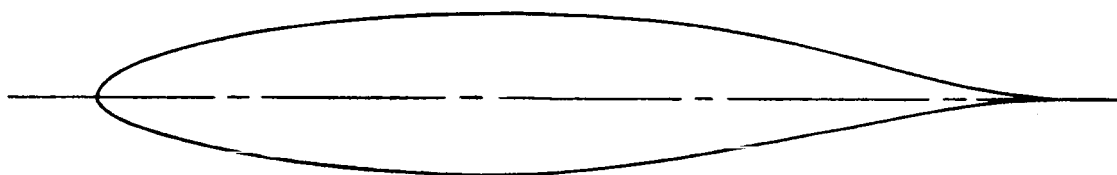32' 8.8"

FIGURE 1.- THREE VIEW DRAWING OF THE TEST AIRPLANE.

Figure 2.– The test airplane as instrumented for test flights.

NACA 66,2X-216
TIP AIRFOIL

NACA 66,2X-116
ROOT AIRFOIL

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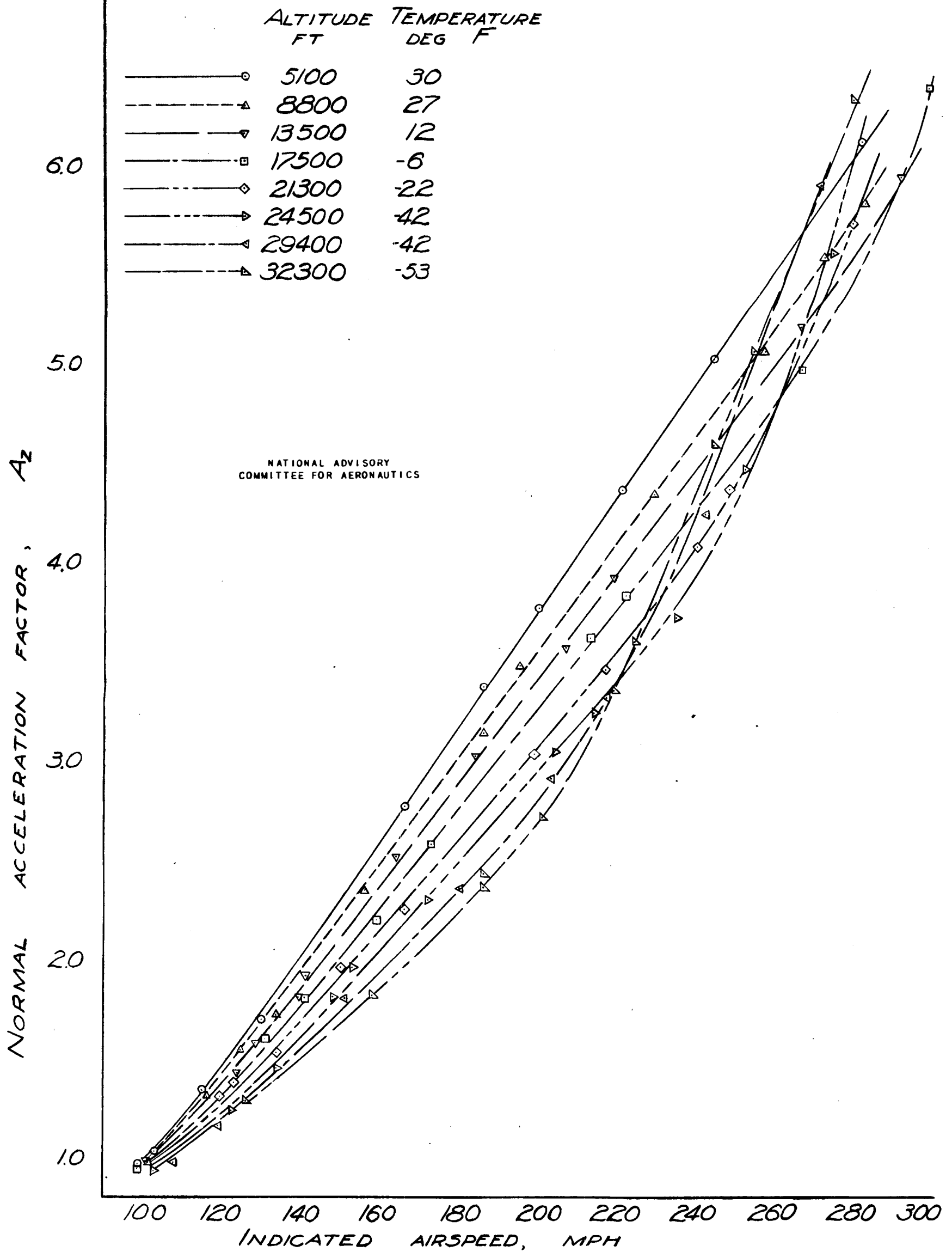FIGURE 3.—PROFILES OF TIP AND ROOT
AIRFOIL SECTIONS OF THE
TEST AIRPLANE

ALTITUDE TEMPERATURE
   FT   DEG F

	ALTITUDE FT	TEMPERATURE DEG F
⊙	5100	30
△	8800	27
▽	13500	12
⊡	17500	-6
◇	21300	-22
▷	24500	-42
◁	29400	-42
△	32300	-53

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NORMAL ACCELERATION FACTOR , A_z

INDICATED AIRSPEED, MPH

FIGURE 4.— MAXIMUM NORMAL ACCELERATION FACTOR OBTAINABLE
IN GRADUAL STALLS PLOTTED AS A FUNCTION OF
INDICATED AIRSPEED FOR VARIOUS ALTITUDES.
TEST AIRPLANE.
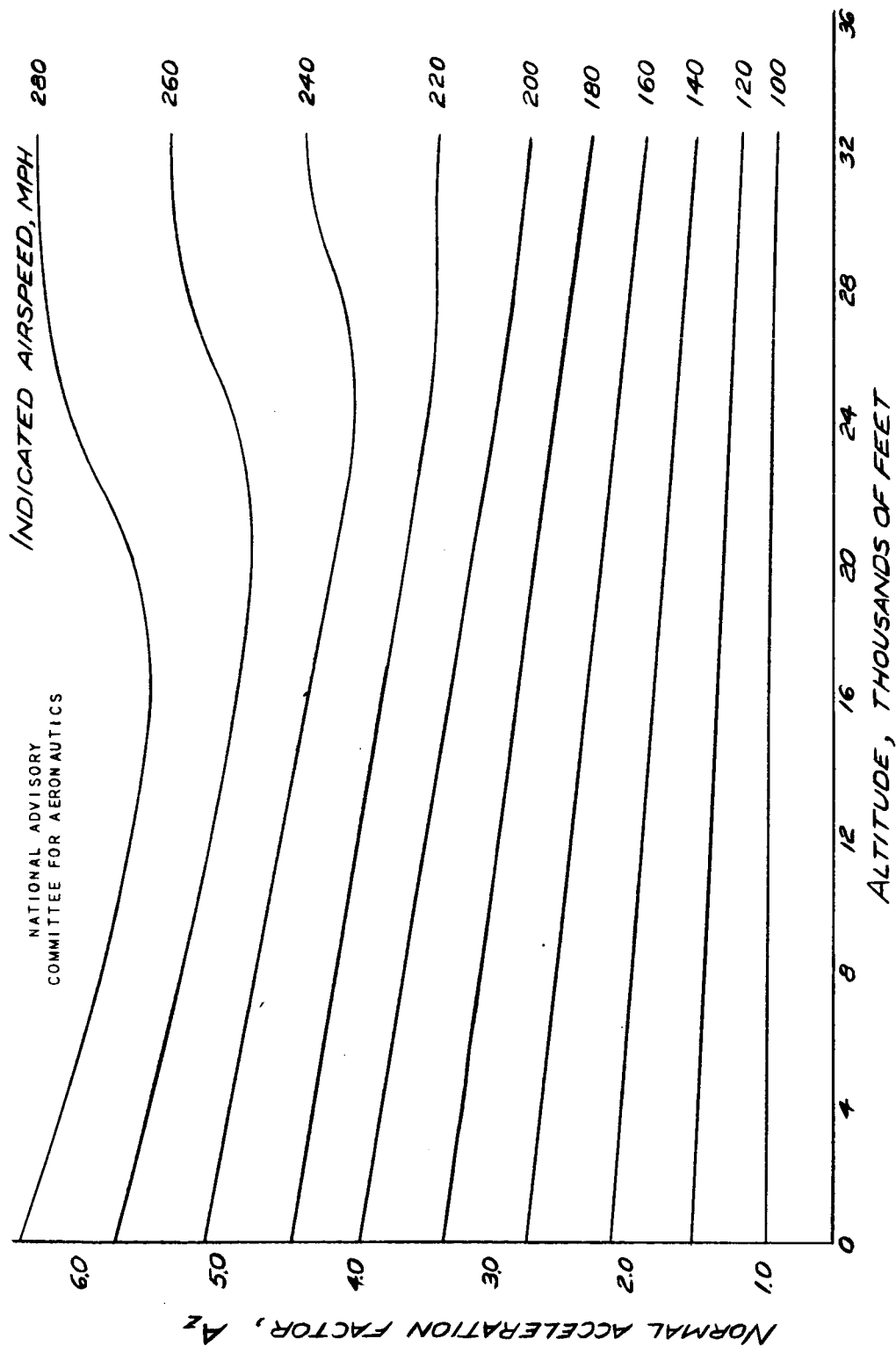
FIGURE 5.— MAXIMUM NORMAL ACCELERATION FACTOR OBTAINABLE IN GRADUAL STALLS PLOTTED AS A FUNCTION OF ALTITUDE FOR VARIOUS AIRSPEEDS. TEST AIRPLANE.
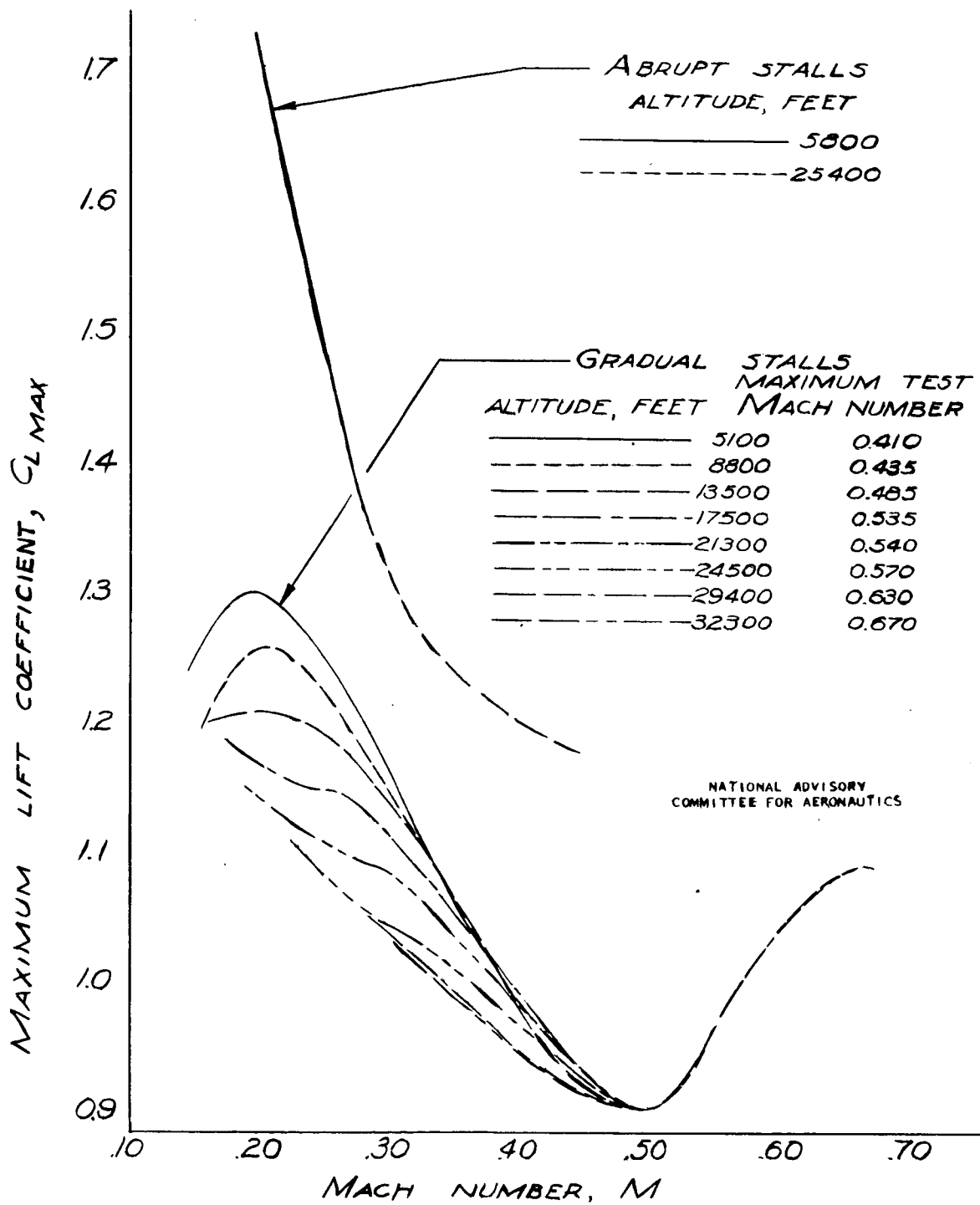
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ABRUPT STALLS
ALTITUDE, FEET

——————————— 5800
— — — — — — — —25400

GRADUAL STALLS
MAXIMUM TEST
ALTITUDE, FEET MACH NUMBER

ALTITUDE, FEET	MACH NUMBER
——————— 5100	0.410
— — — — — 8800	0.435
— — — — 13500	0.485
— — — —17500	0.535
— —— — —21300	0.540
— — — — —24500	0.570
— — — —29400	0.630
— — — —32300	0.670

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MAXIMUM LIFT COEFFICIENT, $C_{L_{MAX}}$

MACH NUMBER, M

FIGURE 6.— MAXIMUM LIFT COEFFICIENT OBTAINABLE IN
GRADUAL AND ABRUPT STALLS PLOTTED
AS A FUNCTION OF MACH NUMBER FOR
VARIOUS ALTITUDES. TEST AIRPLANE.

A-5
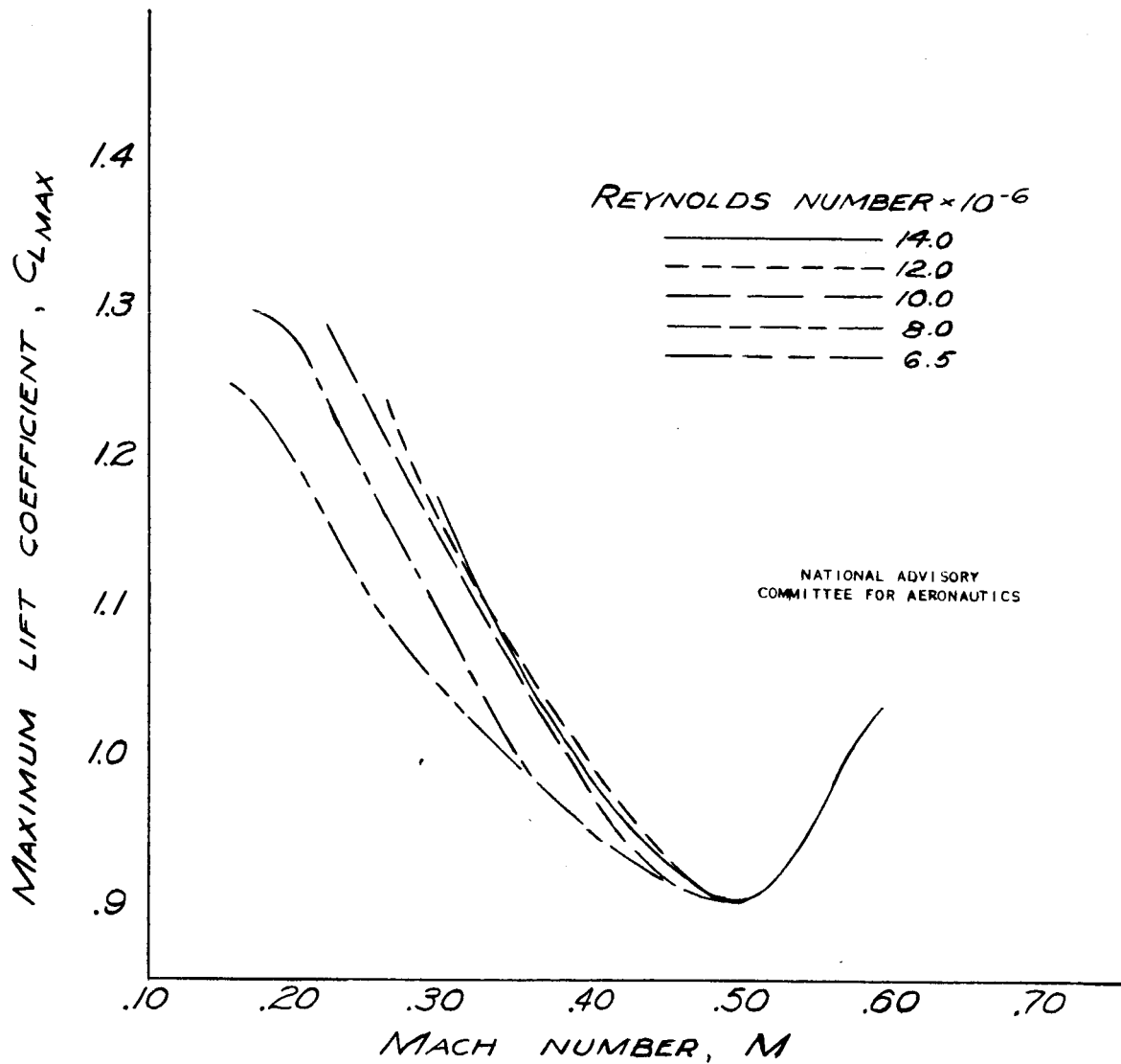
FIGURE 7.— VARIATION OF MAXIMUM LIFT
COEFFICIENT OBTAINABLE IN GRADUAL
STALLS WITH MACH NUMBER FOR
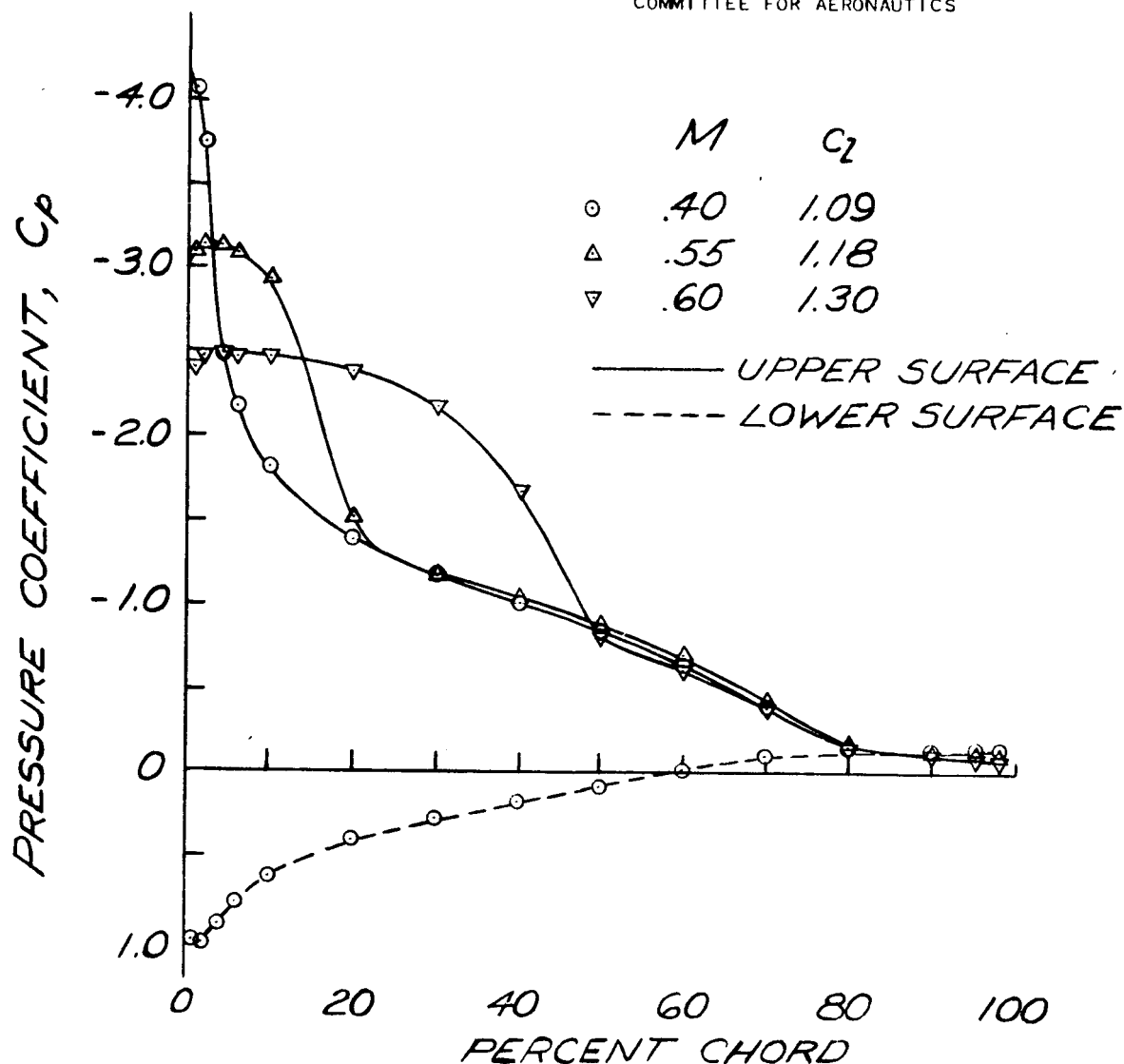VARIOUS REYNOLDS NUMBERS. TEST
AIRPLANE.

A-5

FIGURE 8.- PRESSURE DISTRIBUTION OVER
THE NACA 16-515 AIRFOIL AT AN
ANGLE OF ATTACK OF 11° AT
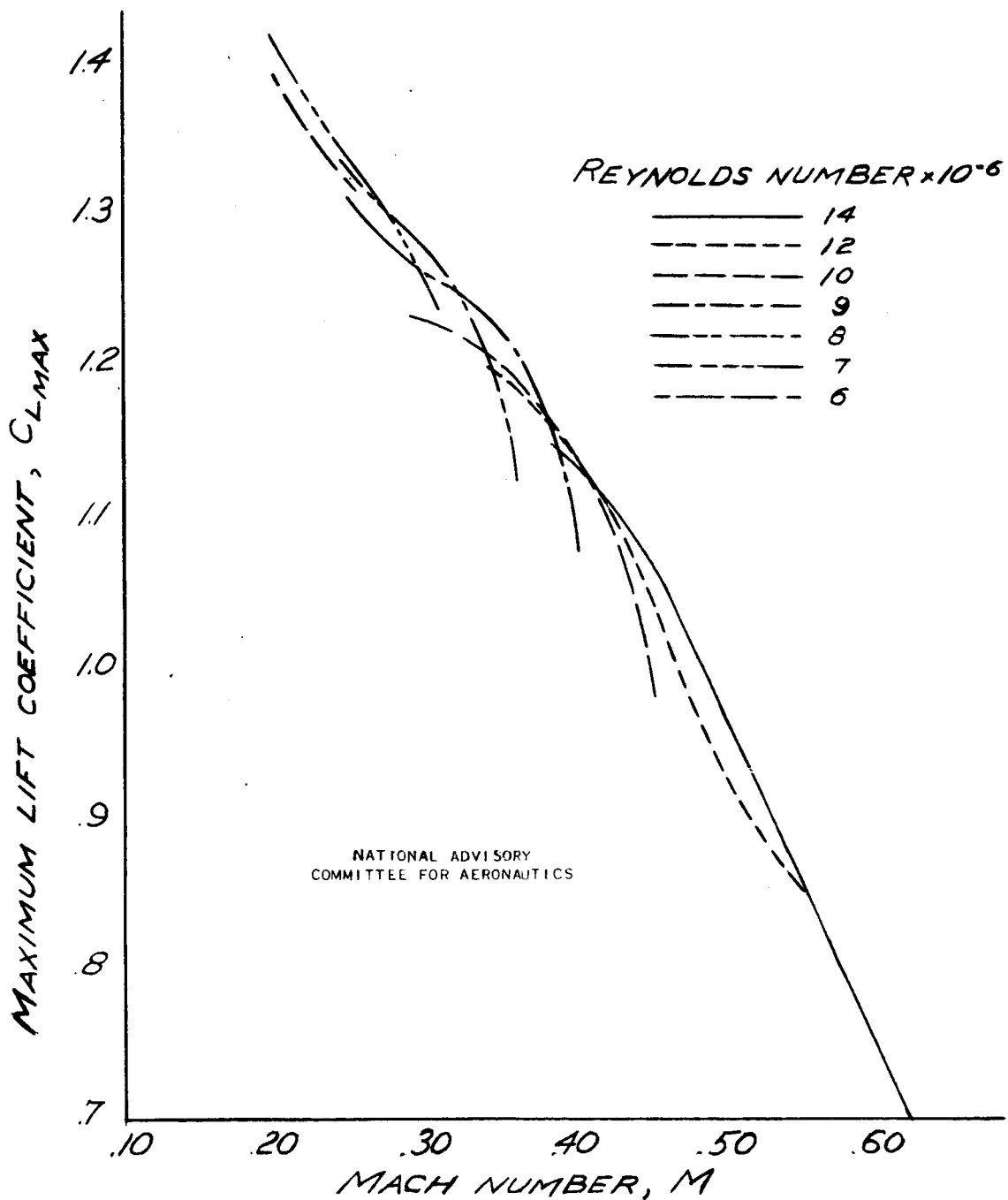MACH NUMBERS OF 0.40, 0.55
AND 0.60. (REFERENCE 3.)

FIGURE 9. — VARIATION OF MAXIMUM LIFT
COEFFICIENT OBTAINABLE IN
GRADUAL STALLS WITH MACH
NUMBER FOR VARIOUS REYNOLDS
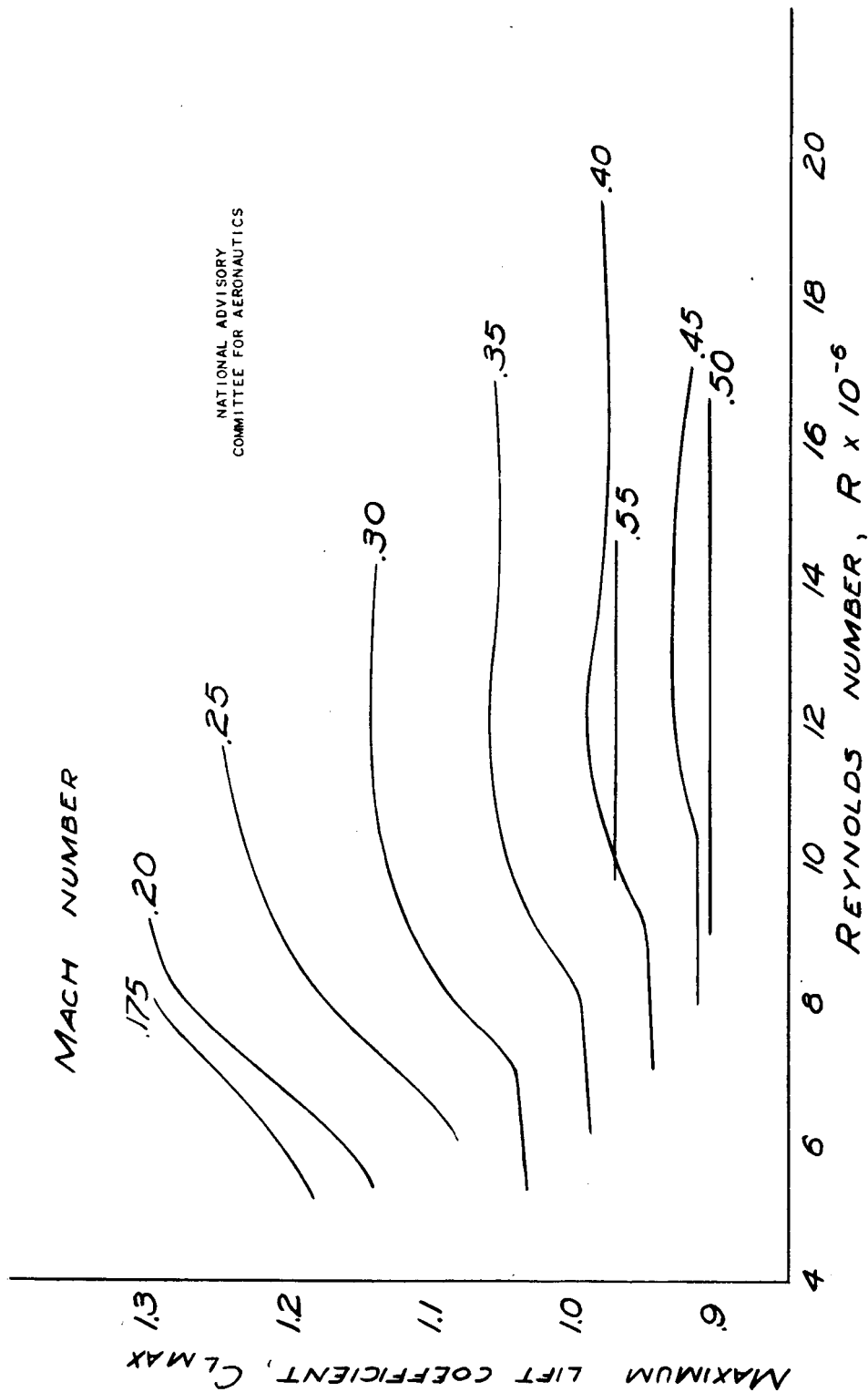NUMBERS. AIRPLANE WITH
CONVENTIONAL WING.
(REFERENCE 1).

FIGURE 10. - VARIATION OF MAXIMUM LIFT COEFFICIENT OBTAINABLE IN GRADUAL STALLS WITH REYNOLDS NUMBER FOR VARIOUS MACH NUMBERS. TEST AIRPLANE.

A-5

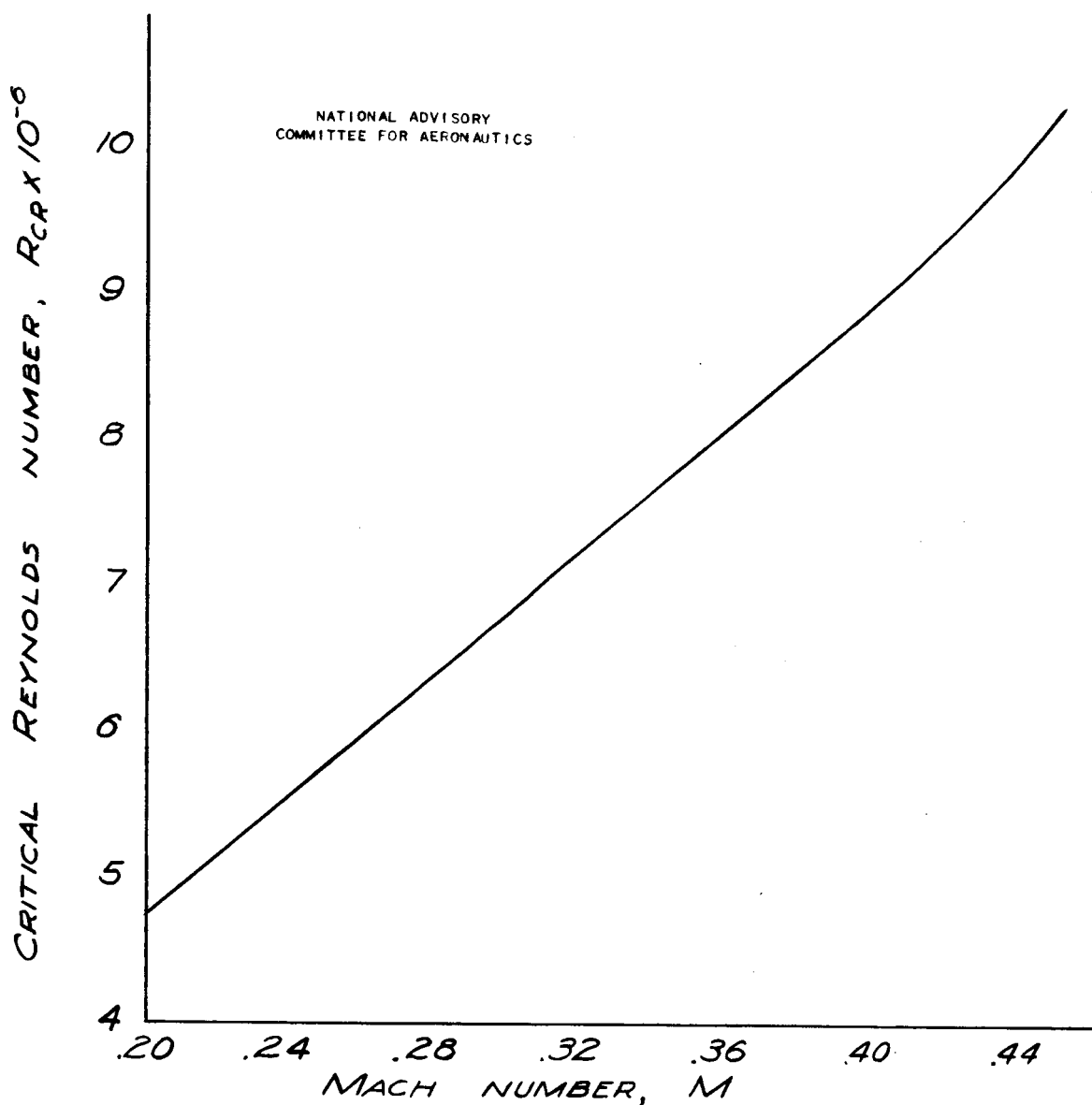

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CRITICAL REYNOLDS NUMBER, $R_{CR} \times 10^{-6}$

MACH NUMBER, M

FIGURE 11.— VARIATION OF CRITICAL REYNOLDS
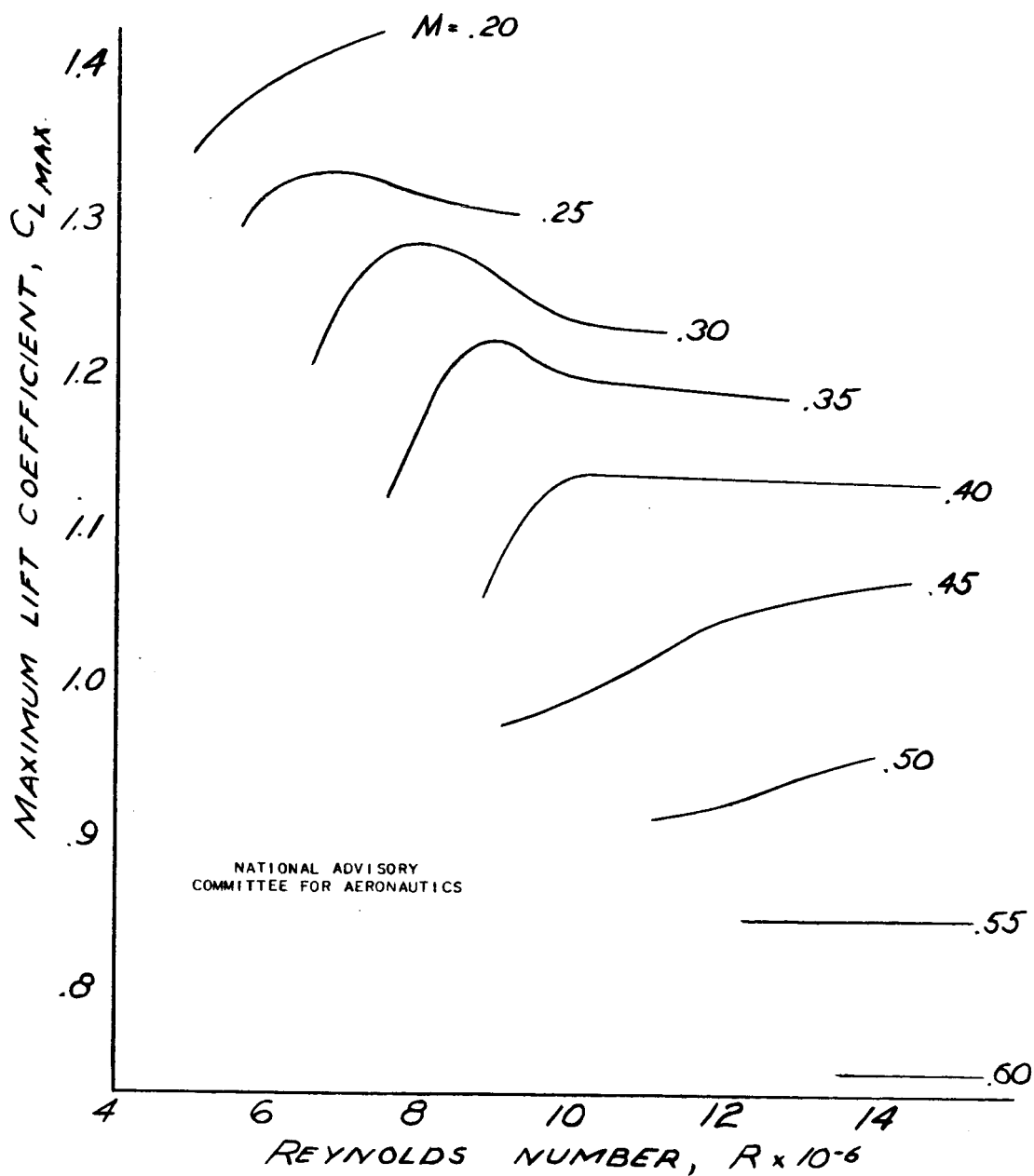NUMBER WITH MACH NUMBER.
TEST AIRPLANE.

A-5

FIGURE 12 — VARIATION OF MAXIMUM LIFT COEFFICIENT OBTAINABLE IN GRADUAL STALLS WITH REYNOLDS NUMBER FOR VARIOUS MACH NUMBERS. AIRPLANE WITH CONVENTIONAL WING. (REFERENCE 1).
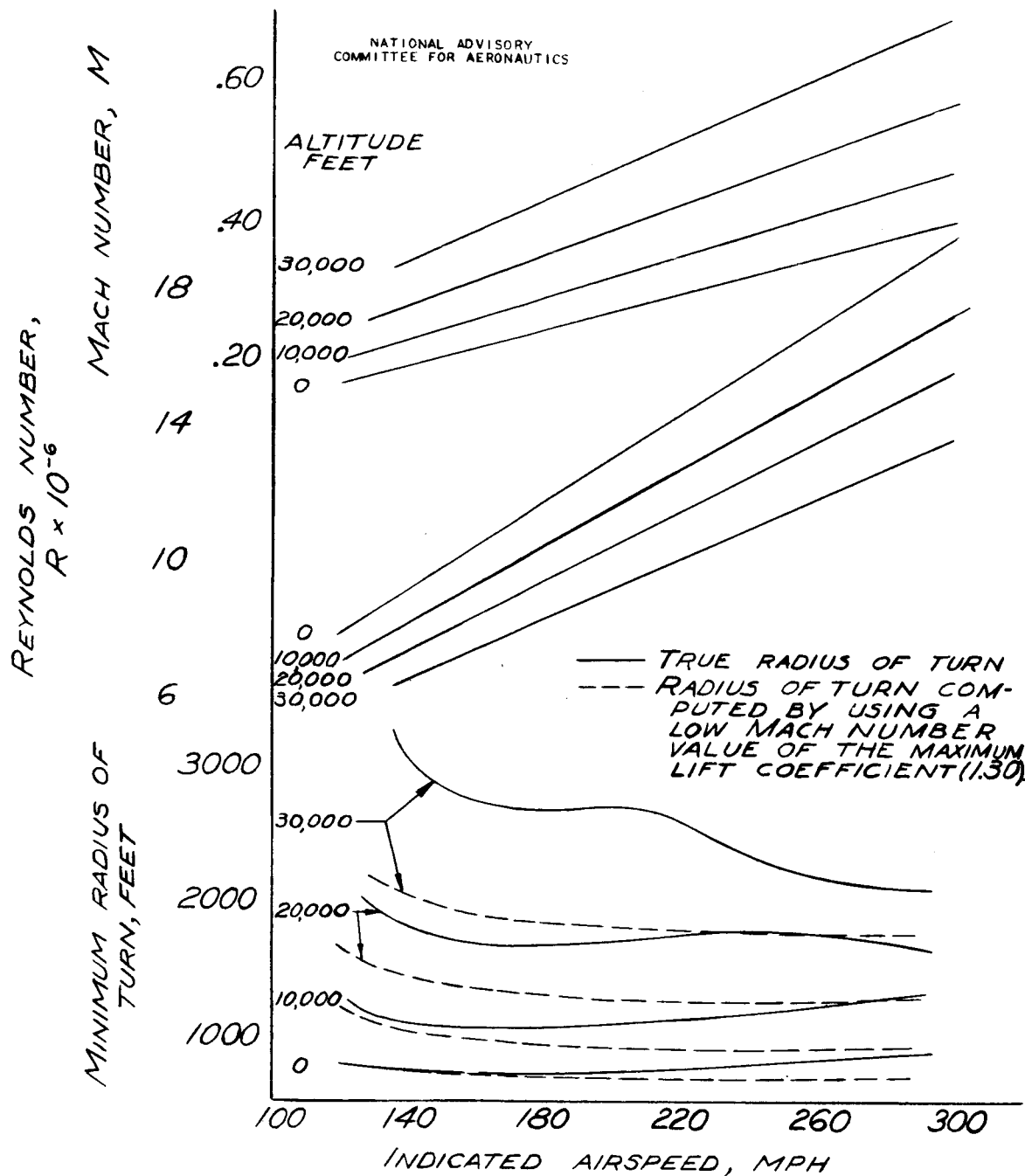
A-5

FIGURE 13. — CALCULATED MINIMUM RADIUS OF
TURN PLOTTED AS A FUNCTION OF
AIRSPEED FOR VARIOUS ALTITUDES.
TEST AIRPLANE.

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	INDICATED AIRSPEED MPH	ALTITUDE FEET	MACH NUMBER	REYNOLDS NUMBER 10^{-6}
⊙	133	5600	.195	8.17
△	169	6000	.250	10.35
▷	130	25200	.283	6.37
☐	196	5800	.286	11.91
◇	168	25400	.365	8.20
△	201	25400	.440	9.92

EXCESS PITCHING ANGLE $\dfrac{\Delta q \bar{c}}{V_T}$, DEGREES PER CHORD LENGTH OF TRAVEL
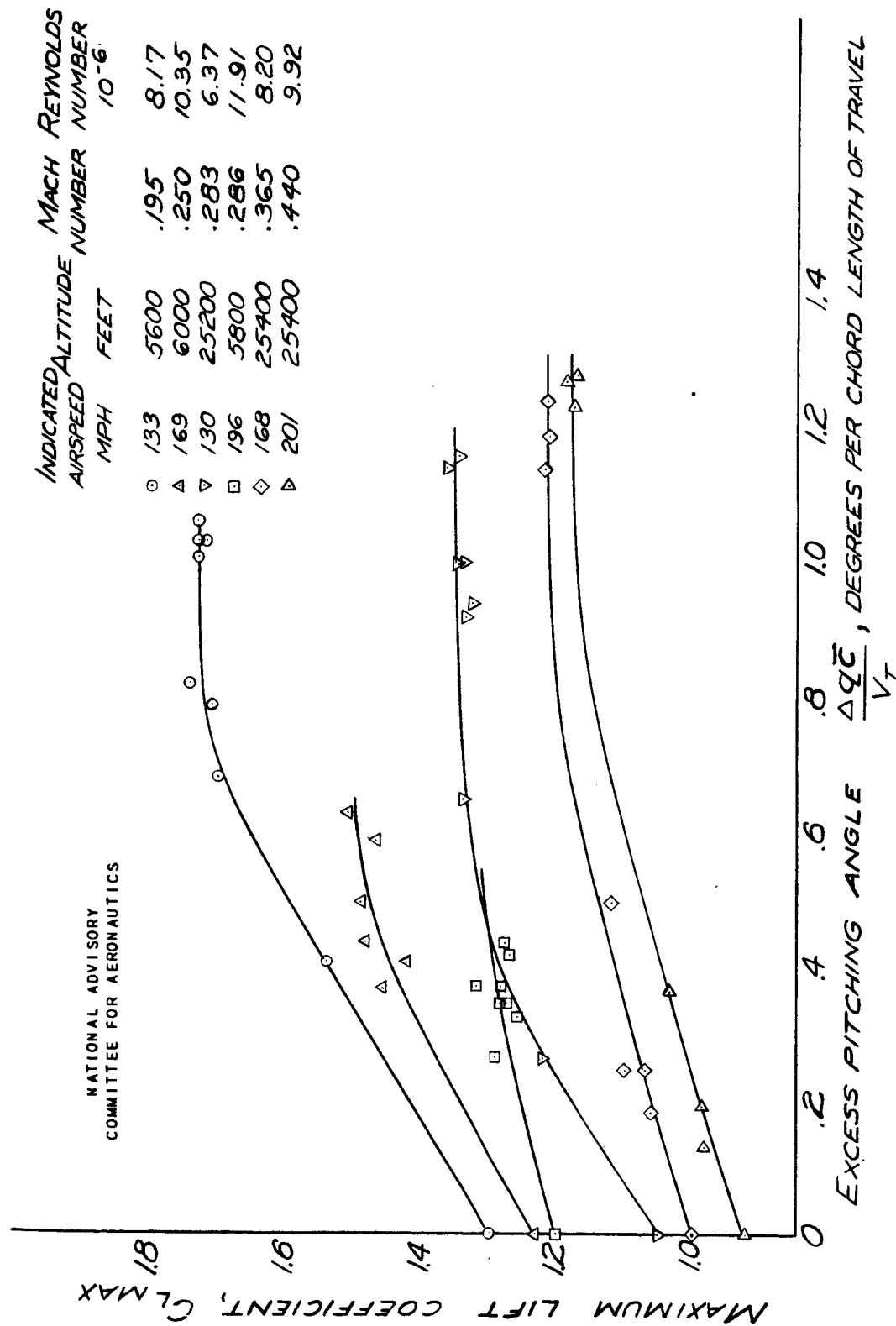
MAXIMUM LIFT COEFFICIENT $C_{L\,MAX}$

FIGURE 14.— MAXIMUM LIFT COEFFICIENT OBTAINABLE IN ABRUPT
STALLS PLOTTED AS A FUNCTION OF EXCESS PITCHING
ANGLE FOR VARIOUS AIRSPEEDS AND ALTITUDES. TEST
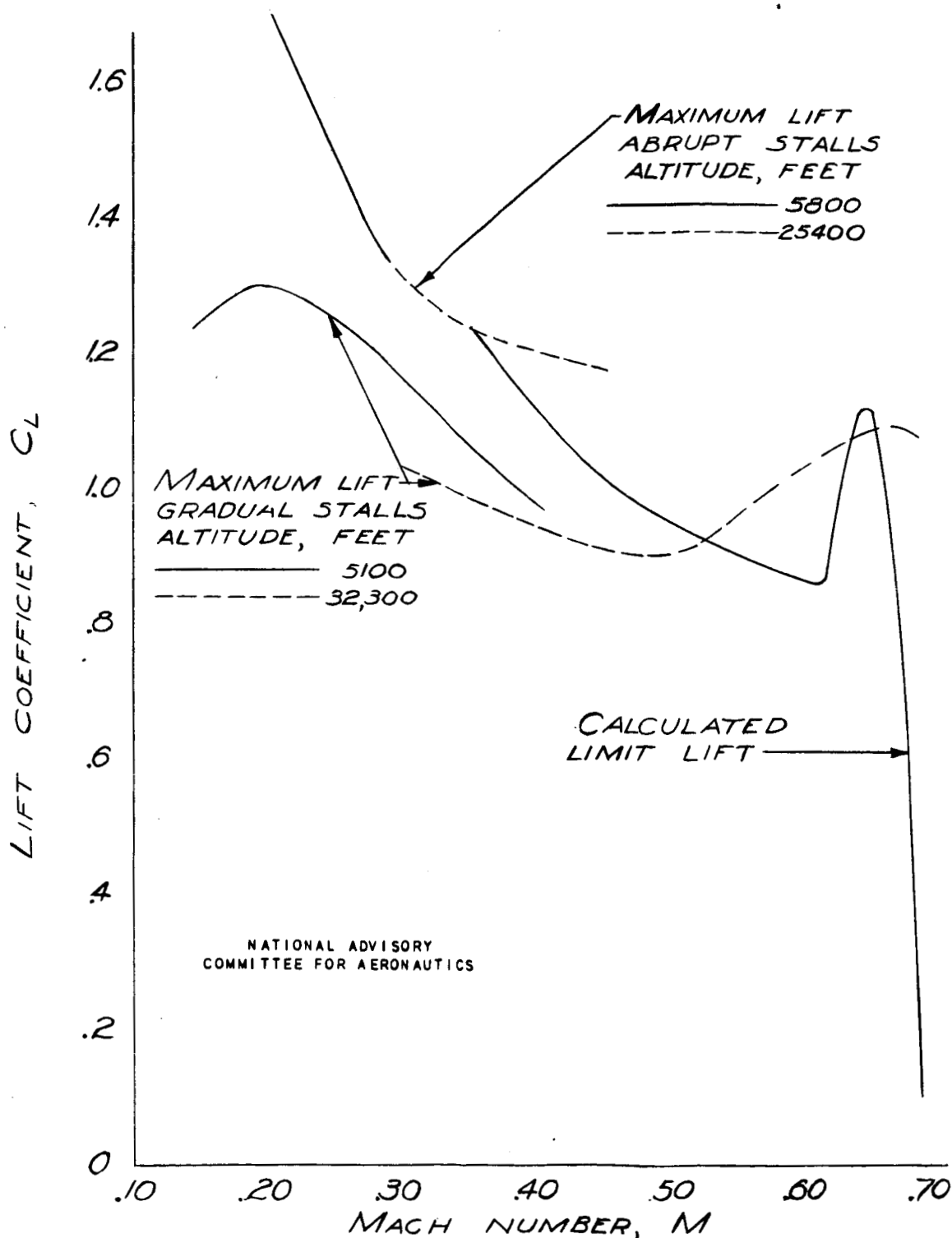AIRPLANE.

FIGURE 15 – MAXIMUM LIFT COEFFICIENT OF THE TEST
AIRPLANE COMPARED WITH CALCULATED
WING LIMIT LIFT COEFFICIENT.